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SCT-1 ROCKET ENGINE

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SCT-1 ROCKET ENGINE

The design for the engine of the SCT-1 rocket is based on past experience with this type of engine, especially with the German V-2's; we therefore decided to use 75% ethyl alcohol as fuel and liquid oxygen as combustion agent. In selecting this fuel, we based our study on what we knew and on what we were trying to achieve in the way of optimum results considering the high exhaust speed of the combustion gases, which in turn of course would give us maximum altitudes. The following table is an analysis of typical fuel characteristics which have been used in experiments with rocket engines.

THEORETICAL EXAUST SPEED OF COMBUSTION GASES IN m/sec			
FUEL	HYDROGEN PEROXIDE	NITRIC ACID	OXYGEN
Hydrogen	4630/3990	4570/4210	5640/5210
Octane	4190/3690	3810/3600	4610/4450
Carbon	3860/3580	3540/3460	4320/4245
Ethyl alchol	3980/3580	3700/3480	4400/4200
Methyl alchol	3900/3480	3640/3360	4245/3990
Aniline	3980/3640	3710/3550	4470/4370
Vinyl ether	3990/3650	3740/3560	4445/4320
Hydrate of hydrazine	3960/3530	3760/3430	4280/3970

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THEORETICAL EXAUST SPEED OF COMBUSTION GASES IN m/sec

FUEL	OZONE	FLOURIDE
Hydrogen	6095/5710	6500/6300
Octane	5090/4930	4920/4820
Carbon	4790/4720	3975/3940
Ethyl alcohol	4840/4650	4750/4620
Methyl alcohol	4640/4420	4650/4480
Aniline	4765/4680	4570/4490
Vinyl ether	4890/4780	4520/4420
Hydrate of hydrazine	4610/4330	5610/5450

The above table is calculated for the same fuels but with combustion agents not yet in actual use.

The highest theoretical exhaust speed in our chemical formulas will be produced by the reaction of pure ozone with pure beryllium at 7,310 m/sec at sea level. By comparison, our high explosives look rather slow; the theoretical exhaust speeds here are:

	m/sec
Nitroglycerine	3,880
Nitrocellulose	3,660
Dynamite	3,300
Double-base powder	3,240
Picric acid	2,600

To burn hydrogen with oxygen, with an excess of hydrogen, we get the following values:

	m/sec
1 kgH2 - 8 kg02	5,170
1 kgH2 - 8 kg02 - 0.5 kgH2	5,030
1 kgH2 - 8 kg02 - 1.0 kgH2	4,890
1 kgH2 - 8 kg02 - 1.5 kgH2	4,770
1 kgH2 - 8 kg02 - 2.0 kgH2	4,680
1 kgH2 - 8 kg02 - 2.5 kgH2	4,570
1 kgH2 - 8 kg02 - 3.0 kgH2	4,470

These data were computed by Dr. Eugen Sanger, a German, and were published in 1950.

Looking at the preceding table, we can see that the solid fuels are those that have the lowest velocity; as Dr. Sanger said, the values here are theoretical and it is really impossible to get these results in a rocket because of incomplete reaction in combustion, heat loss, and theoretical expansion ratios. However, we can say that there is a tendency to reach the values shown in these tables. We can see that the selection of a fuel depends to a great extent on the conditions of the working medium and the achievement of target figures.

CHARACTERISTICS OF V-2 ROCKET ENGINES

Oxygen injection pressure: 26 kg/sq cm (370 lbs/sq inch).

75% alcohol infection pressure: 24.65 kg/sq cm (350 lbs/sq inch).

Pressure in combustion chamber: 15.14 kg/sq cm (215 lbs/sq inch).

Gas extraction velocity at sea level: 2,000 m/sec (6,560 ft/sec).

Temperature in combustion chamber: 2,000° C.

Here are some more data derived from the dimensions of the rocket engine:

To burn 1 kg of 75% alcohol with LOX with the previously mentioned characteristics, we need 6.8 cu dm/1 kg of 75% alcohol in the combustion chamber.

DESIGN CONSIDERATIONS

1. Fuel consumption, 75% alcohol: 1 kg/sec.
   LOX consumption: 1.3 kg/sec.

2. We also assumed that all materials would have to have high resistance and would have to take high temperatures without deformation.

3. The various injectors had to have characteristics of high pulverization and had to be so constituted that combustion would take place in them easily.

4. In order to be able to start on the construction right away, we specified that we were not going to use any material that was not readily available in Mexico.

Our calculations refer to sea level: that is to say, we took an atmospheric pressure of 1.033 kg/sq cm (14.7 [sq inch]/sic/) because in the beginning we though that we would have to make our launch tests at a site near the port of Campeche, since this was an area which offered good environmental conditions.

In accordance with the initial characteristics, the specific impulse had the following value

$$F = \frac{w}{g} V_j \qquad (1)$$

where:

w = weight of combustion gases/sec - 2.3 kg/sec

g = 9.82 m/sec (32.2 ft/sec^2).

V_j = gas escape velocity -2,000 m/sec (such as we know it).

$$F = \frac{2.3}{9.8} \times 2000 = 470 \text{ kg.}$$

Therefore, the specific impulse has the following value:

$$I = \frac{470}{2.3} = 204 \text{ sec.}$$

SHAPE OF COMBUSTION CHAMBER

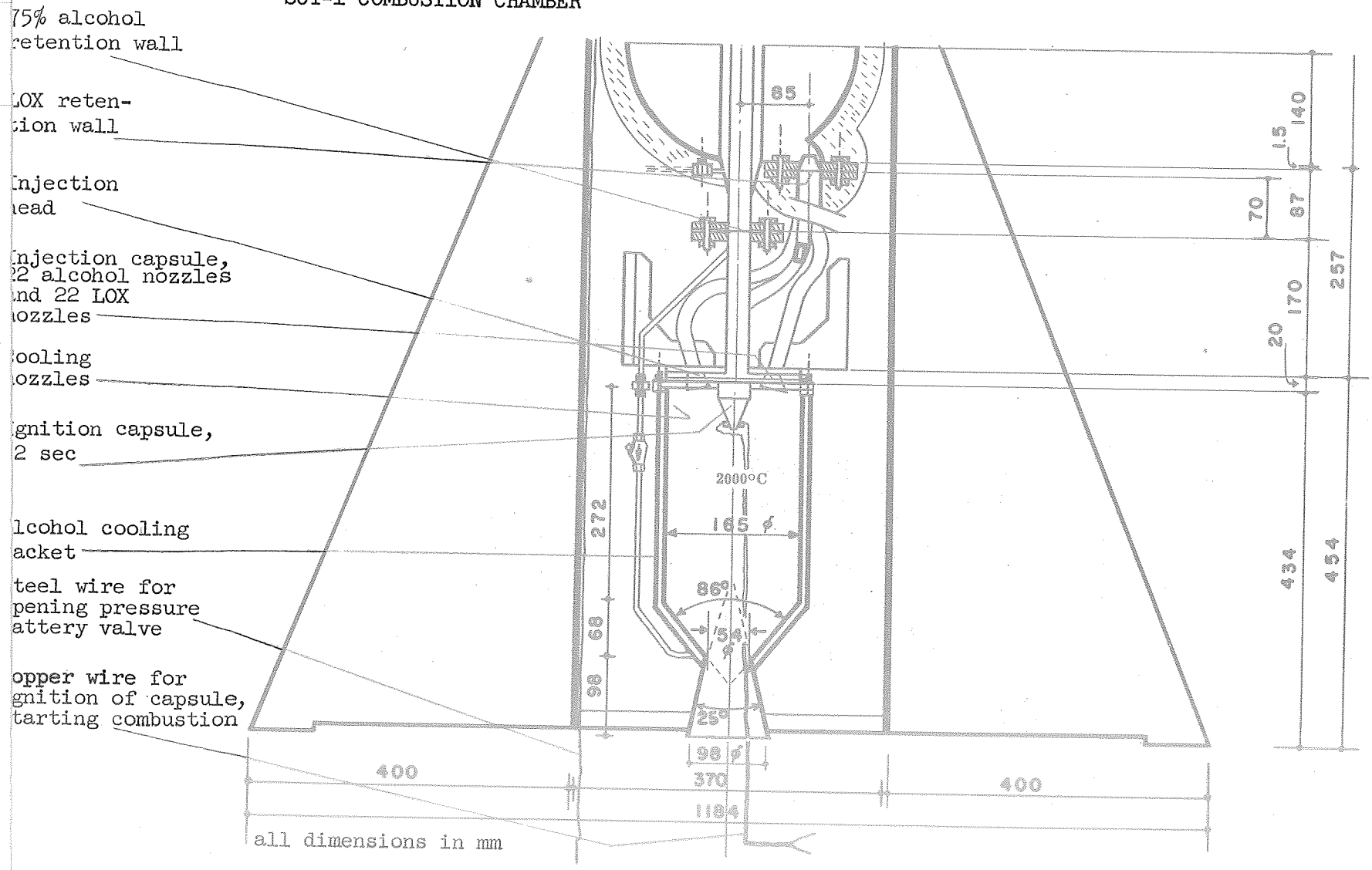
There is no doubt that the shape of the combustion chamber of the V-2's was so designed as to be able to withstand high pressure; that is to say, it had the shape of a sphere which, as we know, can take twice the inside pressure in comparison to the cylindrical shape.

Since the use of the spherical shape would make the entire construction more complicated, we decided to settle on the cylindrical shape here because of its simplicity; here we were also going to get an adequate arrangement for the injection nozzles.

DETERMINATION OF SECTION OF COMBUSTION CHAMBER

The reasonable velocity in the combustion chamber which was indicated here was 60-120 m/sec (200-400 ft/sec), in case of the cylindrical shape; that is to day, these low velocities, with Mach numbers much lower than the

SCT-1 COMBUSTION CHAMBER



exhaust gas velocity, do not produce strong, dynamic vibrations on the inside, which can become quite dangerous; if we were to have to contend with this kind of vibration, we would need high-resistance combustion chambers which in turn would have to be much heavier. In addition, these strong vibrations would lead to low temperatures in the walls of the combustion chambers, as we shall see.

For the calculation of this part, we used the following formulas;

Since we must have continuity of combustion -- which produces a flow of gases -- we have the following:

$$A v_1 = w V_1$$

hence:

$$\boxed{A = \frac{w V_1}{v_1}} \qquad (2)$$

A = section of combustion chamber in sq m or sq ft.

v_1 = inside velocity of combustion gases = 60 m/sec (200 ft/sec).

w = weight of fuel and combustion agent (propellants) in kg/sec or lb/sec = 2.3 kg/sec.

V_1 = specific volume in cu ft/lb or cu m/kg.

From equation (1)

$$w = \frac{Fg}{V_1}$$

Besides, we know that the total volume is =

$$= V_1 = \frac{R T_1}{P_1}$$

But, as we know, the constant of the gases R, due to its molecular weight m, is constant; i.e., Rm = R' = 847 in the metric system.

Substituting in V_1 we have:

$$\boxed{V_1 = \frac{R' T_1}{m P_1}} \qquad (3)$$

Therefore, the circular section of the combustion chamber will have the following value:

$$A = \frac{F g R' T_1}{V_j m P_1 v_1} \qquad (4)$$

Here we have:

$F = 470$ kg

$g = 9.82$ m/sec^2

$R' = Rm = 847 = 1544$ (US system)

$T_1 = 2000^\circ$ C + 273 = 2273° C = 4091.4° F.

V_j = 2000 m/sec which is the velocity of the exhaust gases at sea level and which was then checked out experimentally; further down, we will calculate the coefficients with which the nozzles are working in order to justify the value of V_j, when we compute the nozzle opening.

m = 23.4 in lb/lb mol = 23.4 kg/kg mol.

P_1 = pressure in combustion chamber = 15.1 kg/sq cm = 15.1 X 10^4 kg/sq m = 215 lb/sq inch.

Substituting the values, the section will look like this:

$$A = \frac{470 \times 9.822 \times 847 \times 2273}{2000 \times 23.4 \times 15.1 \times 10^4 \times 60} =$$

$$= 208.85 \times 10^{-4} \text{ m}^2 = 208.85 \text{ cm}^2 .$$

hence:

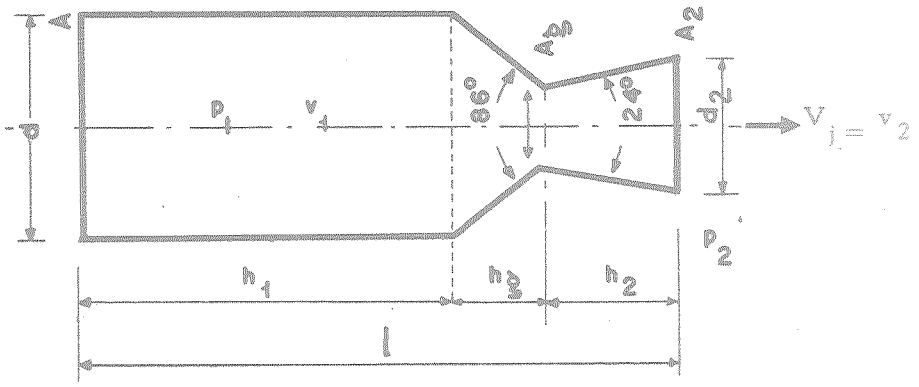
$d = 16.32$ cm.

For the design, we took:

$d = 16.5$ cm.

In order to get the figures we need in order to find the right dimensions for the engine, we set up the following:

$$p_3 = p_2 \qquad V_j = v_2$$



DETERMINATION OF AREA OF THROAT

The effective thrust is smaller than the one computed for an ideal rocket and we can correct it by means of an empirical factor $\triangle$

F = effective thrust

F_i = ideal thrust.

We now have:

$$F = \triangle F_i = \triangle G_F P_1 A_g$$

$$\boxed{A_g = \frac{F}{\triangle C_F P_1}} \qquad (5)$$

Here P_1 = pressure in combustion chamber

A_g = area of throat

C_F = thrust coefficient.

In order to avoid having to go through this whole operation, we can go to any text book on thermodynamics and find the following for the value of C_F:

$$C_F = \sqrt{\frac{2K^2}{K-1}\left(\frac{2}{K+1}\right)^{\frac{(K+1)}{K-1}}\left[1-\left(\frac{p_2}{p_1}\right)^{\frac{K-1}{K}}\right]} + \frac{p_2 - p_3}{p_1}\,\frac{A_2}{A_g} \qquad (6)$$

In this equation:

p_1 = pressure in combustion chamber of nozzle.

p_2 = pressure at nozzle outlet.

p_3 = atmospheric pressure.

It is a good idea to set up $p_2 = p_3$ which means that the formula is reduced solely to the function of the square root; that is to say:

$$\frac{p_2 - p_3}{p_1} \times \frac{A_2}{A_g} = 0$$

For our calculations, we take: $K = 1.22 = \frac{C_p}{C_v}$ because we are dealing here with 75% alcohol with LOX.

$p_1 = 15.1 \times 10^4$ kg/m^2

$p_2 = 1.033 \times 10^4$ kg/m^2 .

Substituting the values in the formula (6), we find that:

$C_F = 1.35$

The correction factor $\triangle$ varies from 0.92 to 1 and depends on the conditions of the nozzle. Since in our case the force we are working with here is figured on an experimentally determined velocity, we have:

$\triangle = 1$

Therefore:

$$A_g = \frac{F}{C_F p_1}$$

Substituting the values, we get:

$$A_g = \frac{470 \text{ kg.}}{1.35 \times 15.1 \times 10^4} = 23.056 \times 10^{-4} \text{ m}^2 = 23.056 \text{ cm}^2.$$

Therefore, the diameter of the throat will be:

$$d_g = 5.4 \text{ cm.}$$

Checking all values and taking 5.4 cm as our definite diameter, the exact surface for the throat is:

$$A_g = 22.85 \text{ cm}^2.$$

We have thus determined the section of the throat in accordance with the thermodynamic characteristics and we now go on to determine the other dimensions of the combustion chamber; here we start on the assumption that we are going to use the characteristics of the combustion chamber of the engines for the V-2; as we said in the beginning, we need an average volume of 6.8 cu dm in order to burn 1 kg of 75% alcohol and the total angle in the cone of the combustion chamber must be 86°.

Through geometric calculations, we determined the form of the combustion chamber in accordance with the dimensions determined earlier; that is to say, a diameter of 16.5 cm and a cylindrical portion with a length of 27.2 cm; the conical part, up to the throat, is 6.8 cm.

We must note here that, if we are going to take into account the recommendations made in various books on the length of the combustion chamber, we would get a chamber that would be 50% smaller volume-wise than the one determined with the data for the V-2; therefore we decided to ignore these recommendations and started designing the entire engine to fit the combustion chamber and nozzles, on the basis of the considerations arising out of the V-2 data.

ESCAPE GAS NOZZLE

The design of the V-2 rocket used an area ratio of 3.3 between the throat and the outlet and had an angle of 24 at the mouth. In accordance with the above data, we have:

$$A_2 = 3.3 \times A_t = 3.3 \times 28.85 = 95.205.$$

$$d_2 = 9.8 \text{ cm.}$$

We did not provide for a cooling chamber in the first engine because we wanted to make an objective determination of the places where we get high temperatures due to the position of the injector capsules.

In the second test we made without cooling jacket on the engine at San Bartolome, we were able to find the places where we were going to have to put cooling nozzles, regardless of whether we were also going to have cooling jackets there.

As we can see in the drawing, we put in four injector capsules, each used for the injection of alcohol in a horizonal jet, as well as for the injection of LOX at an angle of 30° toward the center of the combustion chamber. The final design was very satisfactory because it was possible to get high-temperature flames in the central portion.

The calculations for the alcohol injectors as well as for the LOX injectors -- keeping in mind the drop in the oxygen pressure from 26 to 15.1 kg/sq cm and in alcohol from 24.65 to 15.1 kg/sq cm, using the findings of various authors and figuring the costs -- later on led to completely unacceptable results because in 50% of the cases the expenditures for 1 kg of 75% alcohol and 1.3 kg LOX came out smaller. We are not going into the computations here because we do not really need to do so since the number of holes and diameters were determined experimentally.

The injection system adjustment was a very delicate job because, in addition to handling the flow of liquid we also had to have perfect injection; we achieved this in the alcohol injection nozzles by means of radial guides and in the oxygen nozzles by means of some screws in the conduction pipes. In the chamber we had assemblies of miniature parts; this gave us a perfect mixture in each capsule so that, after testing each motor, we observed practically perfect combustion in each capsule. Number of injectors:

for alcohol: 22 in each capsule; injector diameter 0.8 mm;

for oxygen: 22 in each capsule; injector diameter: 15 injectors with a diameter of 0.9 mm and seven injectors with a diameter of 0.85 mm.

This means that we had a total of 88 alcohol and 88 oxygen injectors.

In the cylindrical portion of the combustion chamber, at a distance of 15 mm, we had 24 cooling nozzles with a diameter of 0.4 mm; our injection pressure was 24.65 kg/sq cm (350 lbs/sq inch); during the second test of the motor we had designed, we observed that the material in this part became red-hot. In the SCT-1 engine, the only cooling occurred when the rocket was launched into space; in the other engines we noticed failures in some of the cooling nozzles; this is why we later on provided the next rocket, that is the SCT-2, with a complete alcohol cooling system; that is to say, the alcohol had to pass through the cooling jackets before arriving at the injection nozzles.

We decided to use inoxidable chrome-nickel steel because it was more resistant at higher temperatures and because we had this material available in the required size and shape in Mexico.

The strongest stress to which the design was subjected was as follows:

(a) in the cylinder of the combustion chamber, with a sheet thickness of 1.6 mm (1/16"):

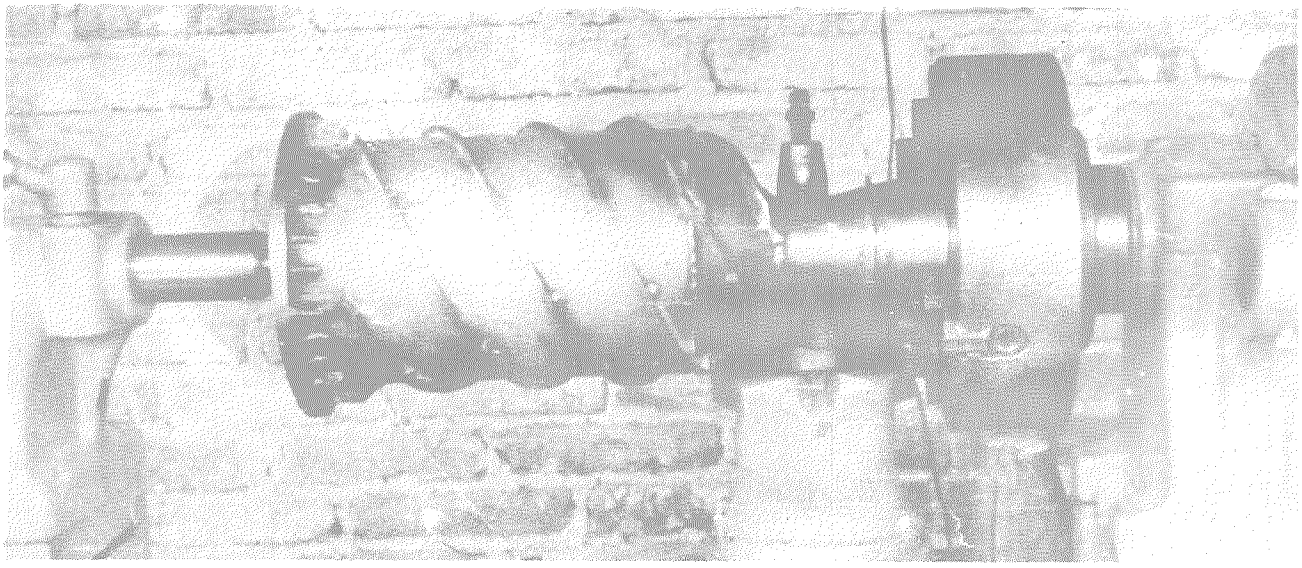
$$f = \frac{d \times p_1}{2e} = \frac{16.5 \times 15.1}{2 \times 0.16} = 778 \text{ kg/cm}^2.$$

(b) in the outside cylinder of the cooling jacket, with a sheet thickness of 1.6 mm:

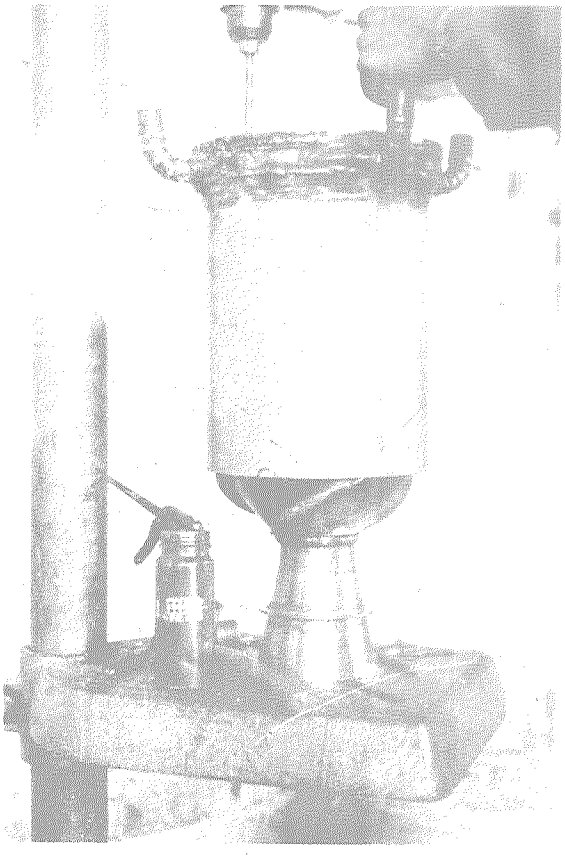
$$f = \frac{d_{ch} \times p_a}{2e} = \frac{18.4 \times 24.65}{2 \times 0.16} = 1417 \text{ kg/cm}^2.$$

These stresses at normal temperature were much lower than 3,500 kg/sq cm, where permanent deformation begins in the chrome-nickel steel (304) N_i 18-20/C_r 8-10.

To make the combustion chamber stronger and more resistant to the high temperatures developed in it, we provided the SCT-1 with reinforcing rings (hoops) to strengthen the cylinder interior as well as the conical part toward the top; but, as we said before, when it was decided to feed all the alcohol through the cooling jacket, engineer Walter C. Buchanan recommended that we insert four conduction screws into the cooling jacket so that each nozzle would have one outlet for each injection capsule.



Combustion chamber of SCT-2 with
4 conducting screws for cooling.



Combustion chamber with upper
cylinder of cooling jacket of
SCT-2.

This system solved our problem here; we can thus say that the design of the SCT-2 embodied all the lessons we had learned.

The ignition capsule shown in the illustration of the engine was designed for a duration of 12 seconds with an adequate mixture of powder and phosphorus; for our next rocket claculations we shall reduce the time since we no longer need this longer interval here. This capsule is ignited electrically and by remote control.

As we can see in the illustrations, the principal alcohol conduit has a retaining wall, as does the outlet duct from the LOX tank; when the pressure battery valve is opened, both of these conduits are closed off and injection into the combustion chamber begins.

Further details of the design can be seen in the engine illustrations.

We have thus shown how we designed the rocket engine. However, it was necessary to adapt its characteristic so that we could work with a 1.4 ratio between LOX and alcohol; that is to say, the chamber worked with a combustion gas weight of 2/4 kg/sec. The dimensions of the tanks and the shape of the rocket were determined on the basis of the V-2 data; this is why the combustion time was 44 seconds.